

NASA TM X-55575

AN ACTIVE AND/OR PASSIVE SPACECRAFT ATTITUDE CONTROL

BY
DENNIS K. McCARTHY

GPO PRICE \$ _____

CFSTI PRICE(S) \$ _____

Hard copy (HC) 2.00Microfiche (MF) .50

ff 653 July 65

MAY 1966

NASA

GODDARD SPACE FLIGHT CENTER

GREENBELT, MARYLAND

N67 11357

(ACCESSION NUMBER)

32

(PAGES)

TMX-55575

(NASA CR OR TMX OR AD NUMBER)

(THRU)

(CODE)

30

(CATEGORY)

AN ACTIVE AND/OR PASSIVE SPACECRAFT
ATTITUDE CONTROL

by

Dennis K. McCarthy

Goddard Space Flight Center
Greenbelt, Maryland

CONTENTS

	<u>Page</u>
SUMMARY	v
INTRODUCTION	1
PASSIVE ATTITUDE CONTROL	2
CAPABILITIES AND LIMITATIONS OF THE PASSIVE ATTITUDE CONTROL	10
ACTIVE ATTITUDE CONTROL	11
Mechanical System Feasibility	12
Sensor Control System	13
Spacecraft Control System	14
Holding to a Target	16
Employing the Passive Attitude Control	16
CONCLUSION	17

LIST OF ILLUSTRATIONS

<u>Figure</u>		<u>Page</u>
1	Basic Structural Configuration	18
2	Paddle Orientation	19
3	Components of Solar Radiation Pressure	20
4	Resultant Solar Radiation Pressure Components	21
5	Components of Solar Radiation Pressure with Respect to the Paddle	22
6	Major Axes with Respect to Each Other	23
7	Spacecraft External Configuration	24
8	Proposed Active Attitude Control	25
9	Simplified Sensor Control System	26
10	Possible Sensor Output Characteristics	26

AN ACTIVE AND/OR PASSIVE SPACECRAFT ATTITUDE CONTROL

by

Dennis K. McCarthy

SUMMARY

This report outlines the feasibility of developing an active attitude control and a passive spacecraft attitude control. It shows the employment of the passive attitude control and its use in conjunction with the active attitude control of a non-spinning spacecraft.

AN ACTIVE AND/OR PASSIVE SPACECRAFT ATTITUDE CONTROL

INTRODUCTION

Attitude stabilization is desired for satellites primarily for maintenance of a fixed orientation with respect to the sun, to obtain maximum utilization of solar cells and to maintain a fixed temperature equilibrium.

To achieve these objectives one or more attitude stabilization systems are required. Each systems purpose is to maintain a constant attitude by:

1. Damping out oscillations about the desired orientation.
2. Providing a restoring torque in the presence of small perturbations.
3. "Dumping" angular momentum which may be imparted to the vehicle by long term torques imposed on the satellite: radiation, magnetic, gravitational and magnetic interaction.

The principal attitude control systems proposed or used to date include:

Flywheels (including spin stabilization as a special case)

Micro thrust rockets

Gravity gradient devices

Aerodynamic stabilizers

Solar sails

Micro thrust rockets become exhausted and the flywheels become saturated in the presence of small but persistent torques, including those due to solar radiation.

The other methods, while admittedly weak and slow acting stabilizers, are non-saturable and may be designed as passive devices. A solar sail's unique feature is its responsiveness to a heliocentric force. Satellite attitude or orbit eccentricity do not greatly affect the restoring torques from a solar sail.

A solar sail employed to orient the spacecraft and dampen oscillations is therefore recommended as a stabilizer for a spacecraft.

PASSIVE ATTITUDE CONTROL

Figure 1 shows the basic outline of a comparative spacecraft with the solar-cell paddles as the intended solar-sail.

Figure 2 shows the solar-cell paddle orientation with the various axes.

X, Y, Z axes are fixed in space through the spacecraft C.G.

ξ, η, ζ axes are euler's nodal axes through the spacecraft C.G.

x, y, z axes are fixed to the spacecraft through its C.G.

β is the angle of the sun-line with the axis.

α is the angle of the solar paddle with the spin axis

ρ is the angle between the sun-line and the plane of the solar paddle,

Therefore,

$$\begin{aligned}\sin \rho = & \sin \psi \sin \beta \cos \alpha \sin \varphi - \cos \varphi \cos \alpha (\cos \theta \cos \psi \sin \beta \\ & + \sin \theta \cos \beta) + \sin \alpha (\sin \theta \cos \psi \sin \beta - \cos \theta \cos \beta)\end{aligned}$$

and

$$\begin{aligned}\cos \rho = & -\cos \alpha \sin \psi \sin \rho \cos \varphi - \cos \alpha \sin \varphi (\sin \theta \cos \beta \\ & + \cos \theta \cos \psi \sin \beta) + \sin \alpha (\cos \theta \cos \beta - \sin \theta \cos \psi \sin \beta)\end{aligned}$$

Figure 5 shows the components of the solar radiation pressure normal and parallel to the surface of the solar paddle.

(Ref. 1)

E = Flux of radiant energy crossing unit area in unit time.

c = Speed of light

r = Reflectivity

P = Normal force

Z = Shear force

R = Resultant force

Only one radiation source (the sun) and no shielding effects between the spacecraft itself and the solar paddles has been considered. Perfect insulation in the paddle and specular reflectivity (glass surface) on the paddle has also been assumed.

Therefore,

$$P = \frac{E}{c} \cos \rho \left[(1 + r) \cos \rho + \frac{2}{3} (1 - r) \right]$$

$$Z = \frac{E}{c} \cos \rho \sin \rho (1 - r)$$

$$R = \sqrt{P^2 + Z^2}$$

$$= \frac{E}{c} \cos \rho \sqrt{4r \cos^2 \rho + \frac{4}{3}(1 - r^2) \cos \rho + \frac{13}{9}(1 + r^2) - \frac{26}{9}r}$$

$$\zeta = \text{Arctag} \left[\frac{\sin \rho (1 + r)}{(1 + r) \cos \rho + \frac{2}{3}(1 - r)} \right] = \text{Arctag} \frac{Z}{P}$$

The average reflectivity of a solar paddle, $r = 0.10$. Therefore,

$$R = \zeta \cos \rho \sqrt{0.4 \cos^2 \rho + 1.32 \cos \rho + 1.17}$$

Where ζ is the momentum flux, $E/c = mc$ on a surface equal to the whole surface of a side of the paddle, A

$$\zeta = A \frac{E}{c} > 0$$

Figure 4 is the resultant solar radiation pressure components with respect to the various previously described axes.

Axes $X, Y, Z; \xi', \eta', \zeta'; \xi, \eta, \zeta; x', y', z'$ and euler angles ψ, θ, φ are positive as shown by the arrows.

Angles $\alpha, \beta, \zeta, \varphi$ are always positive since their absolute values are only important.

Axis S is positive from the spacecraft to the sun.

Components of the force and the resultant force (N, R, S) are positive in the same direction as axes n, s, r.

Figure 3 shows the components of the resultant force with respect to the solar paddle.

Figure 6 shows the major axes with respect to each other.

The angular velocities of the x, y, z axes are:

$$w_x = \dot{\theta} \cos \varphi + \dot{\psi} \sin \theta \sin \varphi$$

$$w_y = \dot{\psi} \sin \theta \cos \varphi - \dot{\theta} \sin \varphi$$

$$w_z = \dot{\psi} \cos \theta + \dot{\phi}$$

The components of the angular momentum are:

$$h_x = I_x w_x = I_x (\dot{\theta} \cos \varphi + \dot{\psi} \sin \theta \sin \varphi)$$

$$h_y = I_y w_y = I_y (\dot{\psi} \sin \theta \cos \varphi - \dot{\theta} \sin \varphi)$$

$$h_z = I_z w_z = I_z (\dot{\psi} \cos \theta + \dot{\phi})$$

The general moment equation is:

$$\vec{M} = \dot{\vec{h}} + \vec{w} \times \vec{h}$$

Therefore,

$$M_x = \dot{h}_x + w_y h_z - w_z h_y$$

$$M_y = \dot{h}_y + w_z h_x - w_x h_z$$

$$M_z = \dot{h}_z + w_x h_y - w_y h_x$$

The moments for each paddle are:

$$M_x = R_z y - R_y z = I_x \dot{w}_x + w_y w_z (I_z - I_y)$$

$$M_y = R_x z - R_z x = I_y \dot{w}_y + w_z w_x (I_x - I_z)$$

$$M_z = R_y x - R_x y = I_z \dot{w}_z + w_x w_y (I_y - I_x)$$

Summing the moments of the forces on the four paddles:

$$\begin{aligned} \Sigma M_x &= \sum_{i=1}^4 (R_{z_i} y_i - R_{y_i} z_i) = I_x \dot{w}_x + w_y w_z (I_z - I_y) \\ &= I_x \frac{d}{dt} (\dot{\theta} \cos \varphi + \dot{\psi} \sin \theta \sin \varphi) \\ &\quad + (I_z - I_y) (\dot{\psi} \sin \theta \cos \theta - \dot{\theta} \sin \varphi) (\dot{\psi} \cos \theta + \dot{\phi}) \end{aligned}$$

$$\begin{aligned} \Sigma M_y &= \sum_{i=1}^4 (R_{x_i} z_i - R_{z_i} x_i) = I_y \dot{w}_y + w_x w_z (I_x - I_z) \\ &= I_y \frac{d}{dt} (\dot{\psi} \sin \theta \cos \varphi - \dot{\theta} \sin \varphi) \\ &\quad + (I_x - I_z) (\dot{\theta} \cos \varphi + \dot{\psi} \sin \theta \sin \varphi) (\dot{\psi} \cos \theta + \dot{\phi}) \end{aligned}$$

$$\begin{aligned} \Sigma M_z &= \sum_{i=1}^4 (R_{y_i} x_i - R_{x_i} y_i) = I_z \dot{w}_z + w_x w_y (I_y - I_x) \\ &= I_z \frac{d}{dt} (\dot{\psi} \cos \theta + \dot{\phi}) + (I_y - I_x) (\dot{\theta} \cos \varphi \\ &\quad + \dot{\psi} \sin \theta \sin \varphi) (\dot{\psi} \sin \theta \cos \varphi - \dot{\theta} \sin \varphi) \end{aligned}$$

If these equations of M_x , M_y , and M_z are substituted in the equations of motion, they can be integrated numerically for each proposed configuration and for various values of sun-line spin axis angle, β , thereby obtaining the resultant movement and orientation of the spacecraft with respect to the sun.

The hypothesis that θ is small can be applied only to the initial moments. If θ tends to increase, then it cannot be considered small after a period of time and the more complete equations must be employed.

The more specific case of the spin axis, Z being in the X-Y plane will be considered.

Therefore,

1. $\psi = 0$, $\cos \psi = 1$ and $\sin \psi = 0$
2. All of the paddles are illuminated from the upper side and the x, y, z axes are transferred to the major axes X, Y, and Z.

Therefore, the moment equations are:

$$\begin{aligned}
 M_x = \zeta [& - 1.80 \sin (\theta + \beta) \cos (\theta + \beta) \cos a (b) \\
 & + 3.59 \sin (\theta + \beta) \cos (\theta + \beta) \sin a (a) \\
 & - 0.80 \sin (\theta + \beta) \cos (\theta + \beta) \sin a \cos a (b \sin a - a \cos a) \\
 & + 1.20 \sin (\theta + \beta) \sin a \cos a (b) \\
 & - 1.70 \sin (\theta + \beta) \cos^2 a (a)]
 \end{aligned}$$

$$\begin{aligned}
 M_y = \zeta \cos \theta [& + 1.80 \sin (\theta + \beta) \cos (\theta + \beta) \cos a (c) \\
 & + 0.80 \sin (\theta + \beta) \cos (\theta + \beta) \sin^2 a \cos a (c) \\
 & + 1.20 \sin (\theta + \beta) \sin a \cos a (c)]
 \end{aligned}$$

$$\begin{aligned}
M_z = M_y \tan \theta + \zeta \cos \theta \{ &+ 1.80 \sin^2 (\theta + \beta) \cos \alpha [c (\sin^2 \varphi - \cos^2 \varphi) \\
&+ 2b \sin \varphi \cos \varphi] + 0.80 \sin^2 (\theta + \beta) (c) \cos^3 \alpha \\
&- 1.60 \cos^2 (\theta + \beta) (c) \sin^2 \alpha \cos \alpha \\
&+ 4.80 \cos (\theta + \beta) (c) \sin \alpha \cos \alpha \}
\end{aligned}$$

From these equations it can be seen that the only moment dependent on φ is M_z .

Integrating M_z along a complete revolution

$$\begin{aligned}
M_z = M_y \tan \theta + \zeta \cos \theta \left\{ &1.80 \frac{1}{2\pi} \sin^2 (\theta + \beta) \cos \alpha \right. \\
&\left[\frac{c}{2} \left| \varphi \right|_0^{2\pi} - \frac{c}{4} \left| \sin 2\varphi \right|_0^{2\pi} - \frac{c}{2} \left| \varphi \right|_0^{2\pi} - \frac{c}{4} \left| \sin 2\varphi \right|_0^{2\pi} \right. \\
&\left. - \frac{b}{2} \left| \cos 2\varphi \right|_0^{2\pi} \right] - 0.8 \sin^2 (\theta + \beta) (c) \cos^3 \alpha \\
&- 1.6 \cos^2 (\theta + \beta) (c) \sin^2 \alpha \cos \alpha \\
&+ 4.8 \cos (\theta + \beta) (c) \sin \alpha \cos \alpha \\
\\
M_z = &[- 0.8 \sin^2 (\theta + \beta) (c) \cos^3 \alpha \\
&- 1.6 \cos^2 (\theta + \beta) (c) \sin^2 \alpha \cos \alpha \\
&+ 4.8 \cos (\theta + \beta) (c) \sin \alpha \cos \alpha] \zeta \cos \theta \\
&+ M_y \tan \theta
\end{aligned}$$

Then the expressions of the moments are:

$$\begin{aligned}
 M_x = \zeta [& - 1.80 \sin (\theta + \beta) \cos (\theta + \beta) (b) \cos a \\
 & + 3.59 \sin (\theta + \beta) \cos (\theta + \beta) (a) \sin a \\
 & - 0.80 \sin (\theta + \beta) \cos (\theta + \beta) \sin a \cos a (b \sin a - a \cos a) \\
 & + 1.20 \sin (\theta + \beta) \sin a \cos a (b) \\
 & - 1.70 \sin (\theta + \beta) (a) \cos^2 a]
 \end{aligned}$$

$$\begin{aligned}
 M_y = \zeta \cos \theta [& 1.80 \sin (\theta + \beta) \cos (\theta + \beta) (c) \cos a \\
 & + 0.80 \sin (\theta + \beta) \cos (\theta + \beta) (c) \sin^2 a \cos a \\
 & + 1.20 \sin (\theta + \beta) (c) \sin a \cos a]
 \end{aligned}$$

$$\begin{aligned}
 M_z = \zeta \{ & \sin \theta [1.80 \sin (\theta + \beta) \cos (\theta + \beta) (c) \cos a \\
 & + 0.80 \sin (\theta + \beta) \cos (\theta + \beta) (c) \sin^2 a \cos a \\
 & + 1.20 \sin (\theta + \beta) (c) \sin a \cos a] \\
 & + \cos \theta [- 0.80 \sin^2 (\theta + \beta) (c) \cos^3 a \\
 & - 1.60 \cos^2 (\theta + \beta) (c) \sin^2 a \cos a \\
 & + 4.80 \cos (\theta + \beta) (c) \sin a \cos a] \}
 \end{aligned}$$

From these equations it can be seen that:

- a. All of the terms of M_y and M_z are multiplied by c . Therefore, when the centerline of the paddles intersect the spin axis, $c = 0$ and M_y and M_z are equal to zero.

- b. M_x is independent of φ and c and it can be seen that it has the ability of aligning the spin axis with the sun, provided that the rotation about the spin axis is zero.

For the care of orientation of the spin axis in the sun direction, in the desired final position,

$$\theta = -\beta, \psi = 0 \text{ (Spin Axis is in the X - Y plane).}$$

This condition is stable if $\ddot{w}_x = 0$ and $\ddot{w}_y = 0$.

In this condition $\cos \psi \sim 1$ and $\sin \psi \sim 0$ and all of the signs are negative because the paddles are illuminated from the upperside.

From the previous moment equations:

$$M_x = 0$$

$$M_y = 0$$

$$M_z = 0$$

It can also be seen that any small perturbation will be damped out ($\beta \sim 0$) provided that there is a favorable moment of inertia ($I_z - I_y > 0$ and $I_z - I_x > 0$).

For positions of the spin axis between the desired final position (Z-axis coincident with sun-line) and an original position

$$0 < (\theta + \beta) < \frac{\pi}{2}$$

then $1 > \sin (\theta + \beta) > 0$ and $1 > \cos (\theta + \beta) > 0$ and $M_x < 0$ which will displace the positive spin axis towards the sun-line, if

$$\cos (\theta + \beta) [-1.80 (b) \cos a + 3.59 (a) \sin a$$

$$- 0.80 \sin a \cos a (b \sin a - a \cos a)]$$

$$+ [1.2 \sin a \cos a (b) - 1.7 (a) \cos^2 a] < 0.$$

where

1. a, b are positive quantities

2. $0 < \alpha < \pi$ by definition

therefore

$$1 > \sin \alpha > 0 \text{ and } 1 > \cos \alpha > -1$$

If $(\theta + \beta)$ tends to diminish, then the limiting values of α depend on the initial $(\theta + \beta)$ for $M_x < 0$. If it is desired that $(\theta + \beta)$ can initially have any value $0 < (\theta + \beta) < \pi/2$, then the left side of the inequality is bigger when $\cos(\theta + \beta) = 1$. In this case, the inequality is always true if

$$\begin{aligned} & -b(1.8 \cos \alpha + 0.8 \sin^2 \alpha \cos \alpha - 1.2 \sin \alpha \cos \alpha) \\ & + a(3.59 \sin \alpha + 0.8 \sin \alpha \cos^2 \alpha - 1.7 \cos^2 \alpha) < 0. \end{aligned}$$

This permits a variation of values in the choice of parameters a, b and α .

If the right side is negative, it would be necessary that the left side be greater in absolute value. This is not possible when $(\theta + \beta) = \pi/2$, because then the left side is zero. Then, only the last inequality applies, being also

$$1.7(a) \cos^2 \alpha - 1.2(b) \sin \alpha \cos \alpha > 0.$$

This still allows a variation in the values of the parameters a, b and α .

CAPABILITIES AND LIMITATIONS OF THE PASSIVE ATTITUDE STABILIZATION AND ORIENTATION

For earth satellites—the earth's shadow will not only cut off the solar pressure for short periods of time but the effect of earthlight will cause a serious periodic variation in the orientation vector. For this reason the use of this type attitude control to provide a stabilization torque for precision applications, such as on orbiting telescopes, is not considered practical.

For other earth satellites, communications, magnetic fields and particles and solar experiments where a fixed orientation is not as critical, this type attitude control can be employed primarily for charging solar cells (will reduce

cost and weight of the solar array by eliminating cells on one side of the paddles or by placing them on the top cover and will boost power by enabling the exposure of more cells directly to the sun). The thermal control will be greatly simplified and more stable by controlling the attitude of a spacecraft.

If additional surface area is desirable, thin aluminized mylar can be attached between paddles to form a web. This will provide an increase in the stabilization torque and provide for more solar cells if required.

This type attitude control is not limited to solar paddle type satellites only. It can be used with other configurations also, for all that would be necessary would be to attach lightweight arms covered by a thin aluminized mylar skin, forming a surface for the solar radiation pressure to apply a stabilizing force. One possibility would be to have a thin skin formed into a truncated cone around the payload and the solar cells mounted on the payload surfaces facing the sun (see Figure 7). Because of its reliability, simplicity and lightweight, this proposed attitude control deserves serious consideration as a method of controlling attitude and damping oscillations in applications arising from certain types of earth satellites and moderate sized interplanetary probes.

Its chief advantages are semi-infinite lifetime, reliability, lightweight and versatility. Its main liabilities include slow response, relatively low torque capabilities, and very low damping capabilities.

Weighing the above considerations, it appears that this attitude control provides minimal stabilization in a simple and passive manner, but is too weak to meet most high performance requirements. This type offers an especially simple method of obtaining maximum efficiency for platform mounting an array of solar cells and should obviate the need to accept the design penalty resulting from an arbitrarily oriented vehicle.

The design and development effort required to implement this proposed attitude control is small and involves no state of the art advances.

This passive attitude control utilizing solar radiation pressure can be optimized by employing a gyroscopic mechanism that is capable of maintaining the spin axis along the sun-line/spin axis.

ACTIVE ATTITUDE CONTROL

The proposed system is shown in Figure 8. This is for a non-spinning satellite. The sun sensor is rotated until it acquires the sun. Upon acquisition of the sun, the second motor is activated to roll the spacecraft over to the sun-line.

The spin axis which is to be aligned with the sun is the axis of symmetry of the vehicle. The sun sensor is mounted on the top of the vehicle. The plane of the sun sensor slit is perpendicular to the axis of the flywheel on the motor in the center of the spacecraft.

To explain the operation of the system assume that the spacecraft has any arbitrary attitude and zero velocity in all three axes. First, motor #2 is turned on. This causes the flywheel (containing motor #1) and the attached sun sensor to rotate until the sun enters the slit in the side of the sun sensor (note that the axis of the flywheel attached to motor #1 is perpendicular to the plane of the sun sensor slit). During rotation of the sun sensor and flywheel combination the spacecraft will rotate in the opposite direction. However, this is not too significant since the desired end result is only to align the spin axis with the sun.

Motor #1 is now turned on. This will cause the spacecraft to rotate about the axis of the flywheel attached to motor #1. Motor #2 remains fixed in position during this maneuver. Hence, the sun sensor does not move relative to the spacecraft during this period. Rotation will stop when the sun is seen by the top of the sun sensor.

With respect to an investigation of the feasibility of this method of attitude control the following questions are of interest:

- a. Is the mechanical system feasible?
- b. What will be the operating characteristics of the sun sensor control system?
- c. What will be the operating characteristics of the spacecraft control system?
- d. Will the system hold a target?

Mechanical System Feasibility

The mechanical system appears to be quite feasible from a gross point of view. Whether it would be feasible for a specific spacecraft application would depend on the specifications for that application. For example, for a given volume, weight and power drain specification for the motor #1/sun sensor/flywheel combination a maximum torquing rate is available. If this maximum rate is satisfactory for the intended application then the system is feasible. Relative to many other forms of attitude control this scheme would probably be quite satisfactory with respect to torquing rate.

In connection with the fabrication of this system particular attention should be paid to the alignment of the sensor and the flywheel. Other than this, no major difficulties are obvious.

Sensor Control System

Considering the present state-of-the-art of sun sensors this system can probably be made to work very well (see Figure 9). Since the moment of inertia of the sun sensor/flywheel combination will probably be much less than the moment of inertia of the spacecraft (both measured with respect to the axis of symmetry of the spacecraft), rotation of motor #2 is unlikely to produce motion of the spacecraft other than a rotation solely about the axis of symmetry (i.e.: little cross-coupling effect should be present).

During the design of the sun sensor and the control system associated with motor #2, consideration must be given to the problem of capture. That is, the sensor must provide sufficient information to the control system such that sun capture can be initially achieved and so that the sensor will be held to the sun continuously after capture. That is, the problem of ending up with a sun sensor control system in which the sensor oscillates across the point of capture without ever converging on this point must be avoided. This will require a sensor with one of perhaps three characteristics. First, with a field of view wide enough such that an overshoot of the sun does not occur during the period between the instant that the sensor first sees the sun and the end of rotation of motor #2, or, second, with a proportional output characteristic within the center of the field of view and also with a sufficiently wide total field of view so that the aforementioned oscillation problem does not occur; or third, with a quantized or multi-level output characteristic such that the control system would tend to hold or drive the system back to the desired position (see Figure 10).

Since the sensor control system would have a loop gain transfer function closely equal to K/s^2 , some form of compensation would be required in the loop in order to achieve stability.

The sensor control system should be designed to keep the sensor locked onto the sun at all times. This is important since the possibility of capture being lost during the period of rotation of motor #1 is quite high if there is cross-coupling between the two axes which lie in the plane which is perpendicular to the spin axis of the spacecraft and if the field of view of the sun sensor is small.

The entire system should be designed so that motor #1 will operate at all times to hold the sensor to the sun. Motor #2 should be capable of operation

only when the sun is entering the sun sensor (and probably only when the sun sensor is within a close tolerance with respect to the center of the capture position).

Spacecraft Control System

The spacecraft control system comprises motor #1, the attached flywheel and associated electronic circuitry. This control system is turned on only after capture has been achieved by the sun sensor. It is shut off if capture is lost or if the sun enters the top of the sun sensor (i.e.: when the spin axis is pointing at the sun). Theoretically, rotation of the flywheel attached to motor #1 will torque the vehicle about a space-fixed line which is coincident with the axis of rotation of the flywheel when motor #1 is first turned on. Whether or not rotation of the spacecraft actually will occur about this axis for a period sufficient to allow the spin axis to be pointed at the sun is one of the most important questions relative to an investigation of feasibility of this system. To seek the answer to this question consider first the case where the sensor control system motor (motor #2) remains off when motor #1 is on. In general, for this case, capture will be lost during operation of motor #1 unless: (i) the axis of rotation of the flywheel lies in a plane which is perpendicular to the axis of symmetry of the vehicle; (ii) this plane also contains two of the principal axes of inertia of the spacecraft; (iii) the cross section of the ellipsoid of inertia which is coincident with this plane is a circle; and (iv) the axis of symmetry of the spacecraft coincides with the remaining axis of inertia. To prove this consider the equations for the angular momentum of the spacecraft:

$$\begin{bmatrix} H_x \\ H_y \\ H_z \end{bmatrix} = \begin{bmatrix} x & -F & -E \\ -F & y & -D \\ -E & -D & z \end{bmatrix} \begin{bmatrix} \omega_x \\ \omega_y \\ \omega_z \end{bmatrix}$$

where the x axis is the axis of rotation of the flywheel, the y axis perpendicular to both the x axis and the axis of symmetry of the spacecraft, and the z axis is orthogonal to the x and y axes; H_i is the momentum of the system about the i axis; and ω_i is the velocity of the system about the i axis ($i = x, y$ or z); A, B and C are the moments of inertia of the spacecraft about the x, y and z axes respectively; and

$$D = \int \int \int \rho yz \, dV$$

$$E = \int \int \int \rho zx \, dV$$

$$F = \int \int \int \rho xy \, dV$$

with ρ = density and dV = infinitesimal volume (i.e.: D , E , and F are the products of inertia of the spacecraft). Capture will be lost unless no cross-coupling is present. This means that D , E and F must equal zero. This will be true if and only if the x , y and z axes coincide with the principal axes of inertia of the spacecraft. Since the x and y axes are capable of rotation with respect to the spacecraft then coincidence of the x and y axes with the axes of inertia at all possible positions of the x and y axes will occur only if the ellipsoid of inertia is a circle in the x - y plane.

If the x , y and z axes are not aligned with the principal axes of inertia of the spacecraft, then pointing the axis of symmetry at the sun will require a series of switching between the sensor control system and the spacecraft control system. The closer the x , y and z axes are to being aligned with the principal axes of inertia the closer the system operation will be to the desired performance of the system.

Next, consider the case where the sensor control system (for purposes of simplification this system is assumed to be a proportional system) is continuously operating to keep the sensor centered on the sun even when motor #2 is on (motor #1 is on only when the sensor "sees" the sun). The system will perform such that the spacecraft will tend to spiral into the final desired attitude (this is assuming the x , y and z axes are not aligned with the principal axes of inertia. If they are, then the spacecraft should move directly into the desired attitude). This will be a result of the sun sensor control system attempting to maintain the flywheel in the same position relative to the spacecraft as it had when motor #1 was first turned on. The exact total effect of this rotation of the flywheel during operation of motor #2 is not too easily visualized. This is because there are several factors that must be considered simultaneously. As motor #2 rotates the flywheel (in an attempt to compensate for the cross-coupling effect) a reverse torque is produced by motor #2 on the spacecraft. In addition, one should consider the fact that the flywheel is spinning while being rotated. However, the gross effect of all of these elements will probably be as described above.

Regardless of whether motor #2 can be operated independently of motor #1, a design of this system must take into account the necessity of the spacecraft control system to stop after the sun enters the field of view of the top sun sensor and before the spacecraft travels completely through this field of view. A system could be designed to do this. The main design trade-off here would be between motor stopping time and torquing rate of the spacecraft.

Holding to a Target

If there are no extraneous torques acting on the system then the system should quite easily hold the desired target. However, this is a trivial case. If the system does drift out of the field of view of the top of the sensor, then alignment of the axis of symmetry with the sun can be brought about only by recapturing the sun in the side slit of the sun sensor and then reactivating the flywheel of the spacecraft control system.

To this point in the investigation the control problem has been considered only as a two-point problem with zero velocity at both the initial and final points. However, such may not be the case, particularly if the system is attempting to recover after having drifted away from the final capture position. Therefore, this aspect of system operation will now be considered.

Recapture of the desired final attitude is possible only if the sun sensor side slit can be kept aligned with the sun until the top of the sun sensor can be realigned with the sun. For retention of this alignment the sensor control system must have a tracking capability.

To briefly summarize, this system can be used to hold the spacecraft to a target. However, it appears that good performance in this area would require that the rate of system recovery be at least one order of magnitude than the largest expected drift rate.

Employing the Passive Attitude Control

Once the axis of symmetry is aligned with the sun some other control scheme could be used to hold the system in the desired attitude. Whether solar paddles would suffice for this purpose would depend on the requirements of the payload and the ratio of solar paddle restoring torque to imparted torque. That is to say, assuming that a solar paddle control system would work in its own right, its effectiveness as a secondary control system will depend on the response time of the system (e.g.: the maximum restoring torques of the solar paddles, the

dynamic characteristics of the spacecraft, the magnitudes of the disturbing perturbations and the magnitude of the field of view of the sun sensor).

CONCLUSION

The solar paddles, therefore, bring the spacecraft into approximate alignment with the sun-line and then the gyroscopic mechanism optimizes the attitude by maintaining the spacecraft/sun-line axis during the spacecraft's lifetime.

The passive/active attitude control are both applying torques to align the Z-axis with the sun. Upon acquisition of the sun, the active control will shut off, but the momentum of the passive system will overtravel the sun-line, thus re-activating the sun sensor to reacquire the sun, and thus applying an opposite torque which will first stop the overtravel and then re-orient along the sun-line. (Figure 8)

This motion will occur several times until all of the motion has been damped out.

From then on the gyroscopic mechanism corrects and maintains the spacecraft sun-line axis with added assistance from solar radiation pressure.

The proposed attitude control scheme is conceived mainly as a system for changing the attitude of the spacecraft from an initial position to a final position when the values of spacecraft velocity and acceleration are zero at both the initial and final positions. As such, with careful design of the sensor, the sensor control system and the flywheel control system, the system should function quite well.

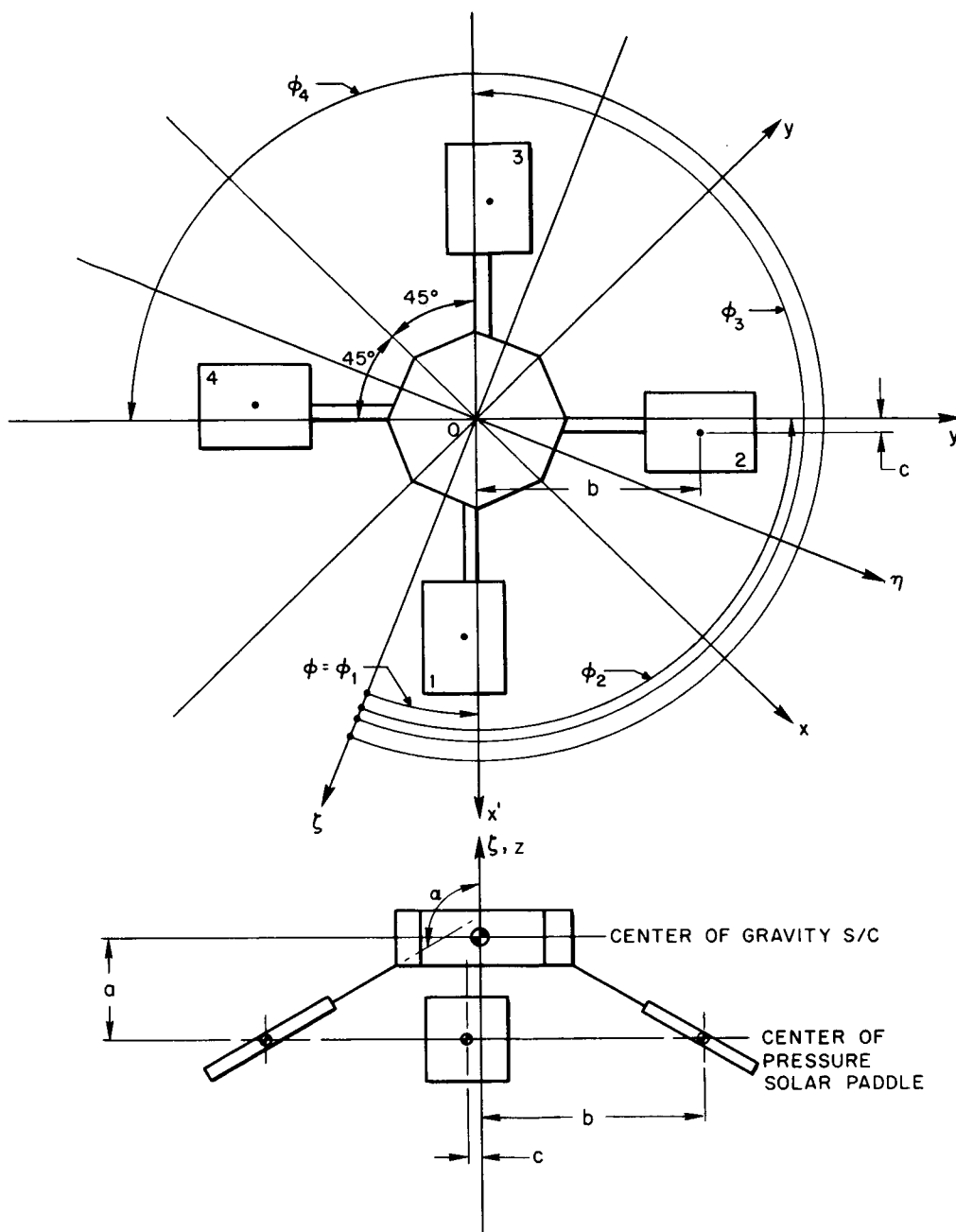


Figure 1—Basic Structural Configuration

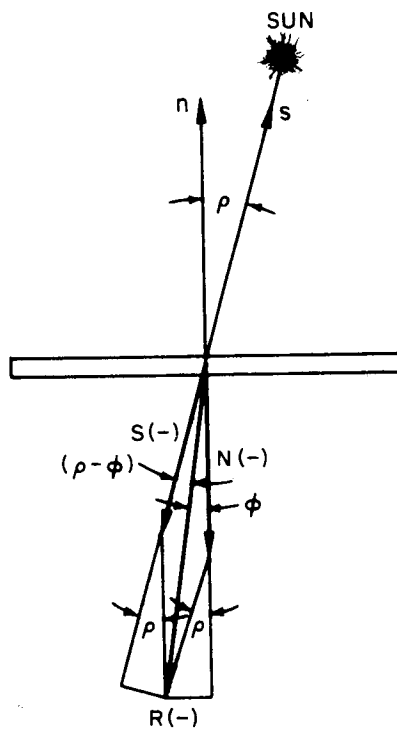


Figure 3—Components of Solar Radiation Pressure



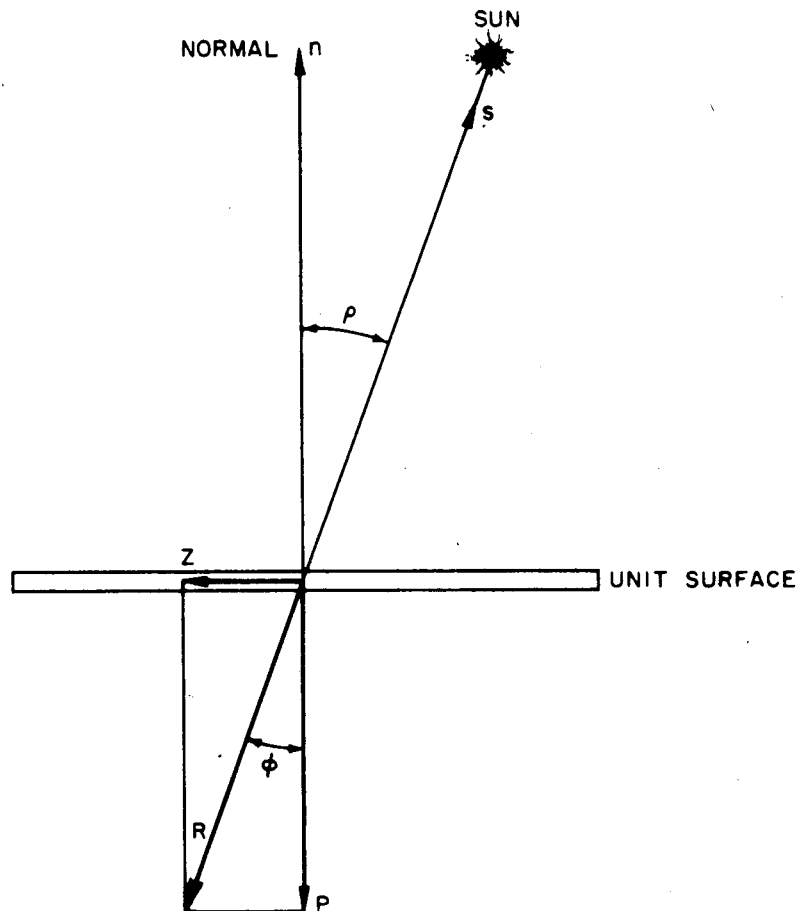


Figure 5—Components of Solar Radiation Pressure With Respect to the Paddle

23

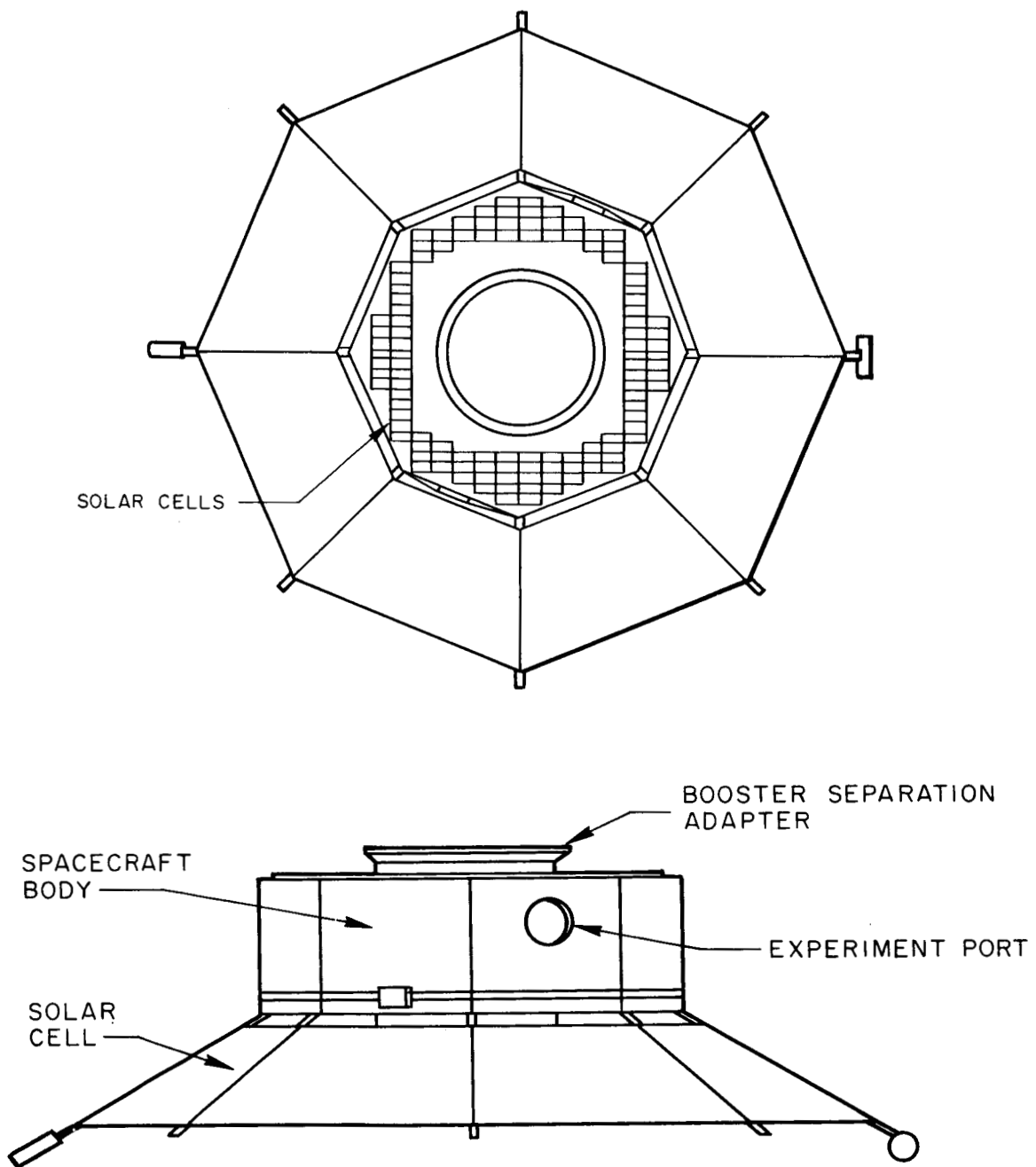


Figure 7—Spacecraft External Configuration

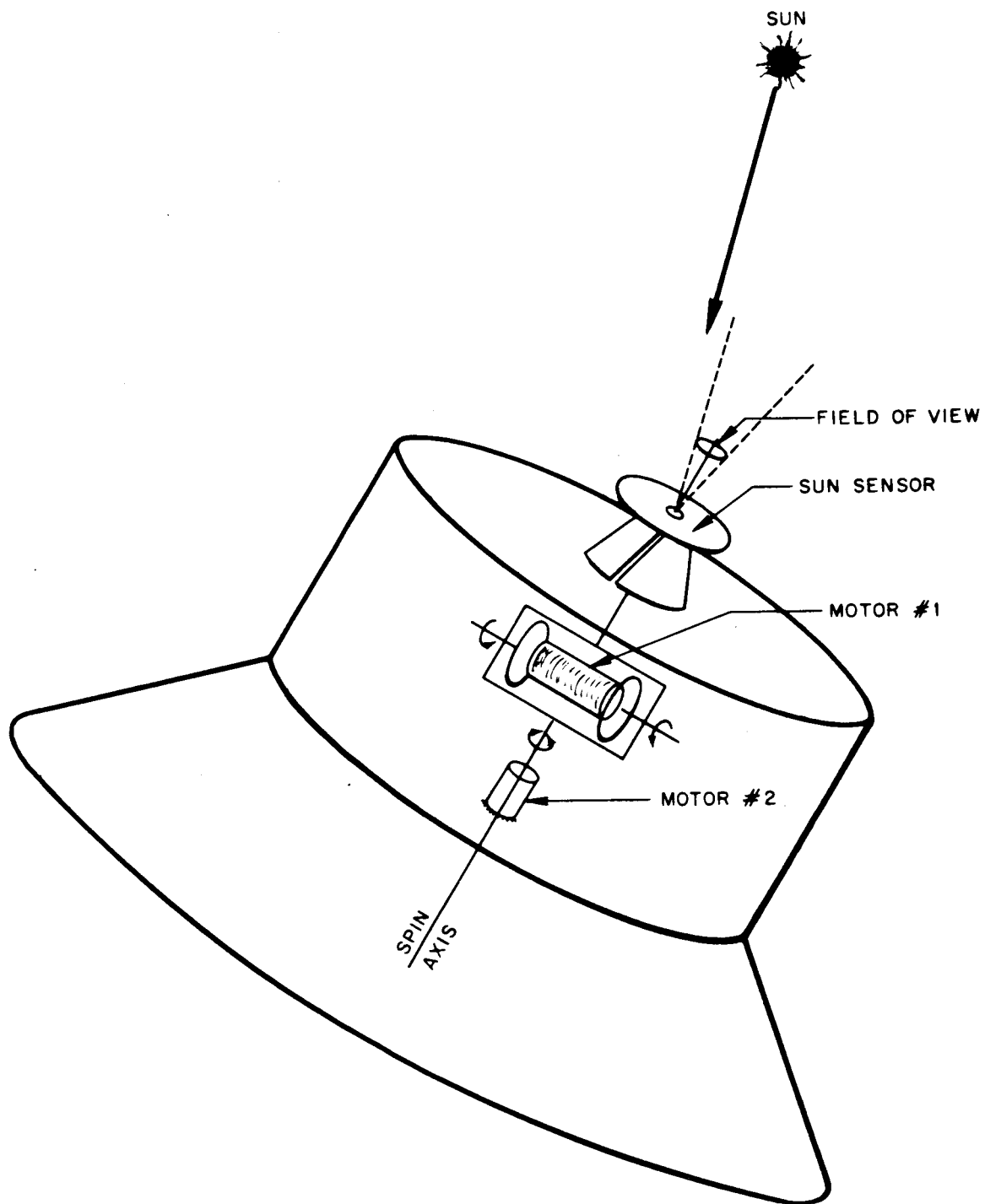


Figure 8--Proposed Active Attitude Control

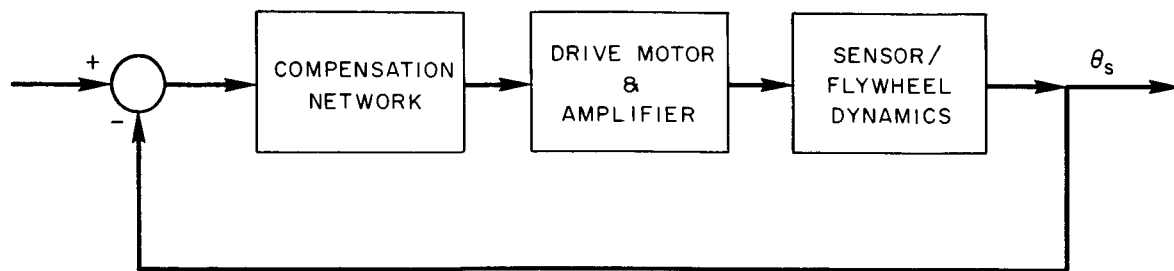


Figure 9—Simplified Sensor Control System

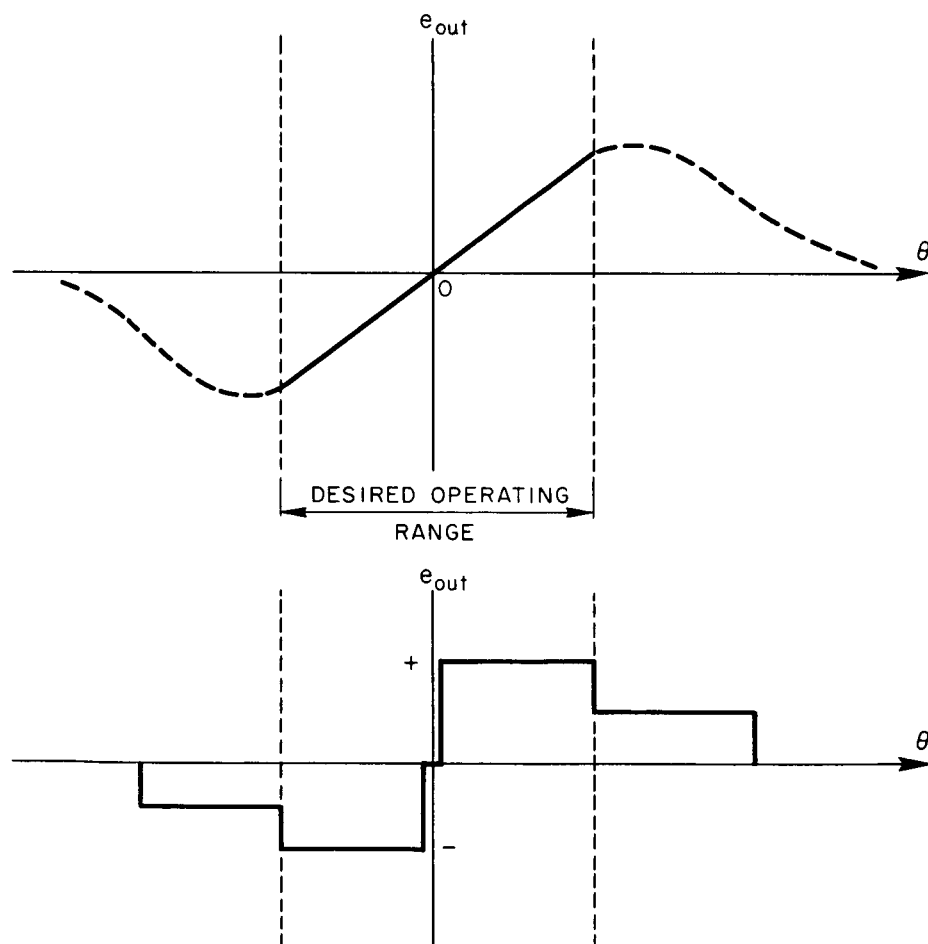


Figure 10—Possible Sensor Output Characteristics